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LOW SPEED, HARD OBJECT IMPACT ON THICK GRAPHITE/EPOXY PLATES

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UNCLASSIFIED TITLE

LOW SPEED, HARD OBJECT IMPACT ON THICK GRAPHITE/EPOXY PLATES.

ABSTRACT

(U) RESULTS OF A PRELIMINARY TEST PROGRAM CONDUCTED TO ASSESS THE EFFE
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OMPRESSION AND COMPRESSION FATIGUE TESTS WERE CONDUCTED ON 42 AND 48 PL
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GE HAD NO SIGNIFICANT EFFECTON ON FATIGUE LIFE AS LONG AS THE MAXIMUM C
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INDEX TERMS ASSIGNED

COMPRESSION
FATIGUE LIFE
DAMAGE
LOW VELOCITY
AIRFRAMES
LAMINATES
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SMALL TOOLS
IMPACT
AIRCRAFT
TEST METHODS
THICKNESS

TERMS NOT FOUND ON NLDB

COMPRESSION FATIGUE TESTS
IMPACT THREATS
STATIC COMPRESSION
THICK GRAPHITE EPOXY PLATES
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RAPHITE/EPOXY PLATES.

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WERE CONDUCTED ON 42 AND 48 PLY AS/3501-6 GRAPHITE/EPOXY PLATES WHICH HAD BEEN
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INDEX TERMS ASSIGNED
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TERMS NOT FOUND ON NLDB
HARD OBJECT IMPACT
LAMINATE FATIGUE STRENGTH
THICK GRAPHITE EPOXY LAMINATES
3501-6 GRAPHITE EPOXY PLATES

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Results of a preliminary test program conducted to assess the effect of low speed, hard object impact (such as dropped hand tools) on thick graphite/epoxy laminates representative of primary aircraft structure are presented. Static compression and compression fatigue tests were conducted on 42 and 48 ply AS/3501-6 graphite/epoxy plates which had been impact damaged by dropped weights. Results show thick laminates are susceptible to damage under impact threats. Laminate fatigue strength can be significantly

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20. degraded by impact damage which is not visible. However, subvisual impact damage had no significant effect on fatigue life as long as the maximum compression strains were held below .0025.

TABLE OF CONTENTS

INTRODUCTION

The high specific strength and stiffness of advanced composite materials makes them attractive candidates for aircraft applications. Navy aircraft currently under development, such as F-18 and AV-8B, use graphite/epoxy composite material in primary structures, such as wings, tails, etc. Naval aircraft are operated under high loads and harsh service environments. Materials used in structural applications must be able to provide reliable performance with minimum maintenance under these conditions. Included in the environmental threats are low speed impacts such as accidentally dropped hand tools, runway stones, and hail.

Tests conducted on specimens of thin graphite/epoxy laminates (8 to 12 plies) and composite faced honeycomb sandwich, references (a) and (b), have shown that mechanical strength, particularly compressive strength, can be significantly degraded as a result of low velocity, transverse normal impact. Because of this potential strength reduction, this exploratory program was initiated to assess the effect of low speed, hard object impact on the thick laminates (42 and 48 plies) currently proposed for use in primary structure on future Navy aircraft.

TEST SPECIMENS

The graphite/epoxy material used in this study and the material called for in the current Navy designs is Hercules AS/3501-6. Two layup constructions representative of primary structure were selected for study. These are:

Laminate A: 48 ply $(\pm 45/0_2/\pm 45/0_2/\pm 45/0/90)_{2S}$

Laminate B: 42 ply $(+45/90/-45/+22.5/-67.5/-22.5/+67.5/\pm 45/+67.5/$
$+22.5/-67.5/-22.5/\pm 67.5/\pm 22.5/0_2/\pm 22.5)_S$

The mechanical properties of these laminates are presented in table I. B laminates were made from prepreg material of 0.0060 ply thickness while A laminates were fabricated from the normal 0.0052 ply thickness material.

TABLE I

LAMINATE PROPERTIES

	Laminate A	Laminate B
E_x	10.2×10^6 psi (70.3 GPa)	8.3×10^6 psi (57.2 GPa)
Thickness	0.272 inch (6.91 mm)	0.254 inch (6.45 mm)

Individual test specimens, 4.0 inches (102 mm) wide by 6.0 inches (152 mm) long, were machined from a large panel of each layup sequence which had been autoclave cured following the material manufacturer's suggested cure cycle.

Six specimens of laminate A and 12 specimens of laminate B were made. Specimens were then inspected both visually and by ultrasonic C-scan prior to testing to insure soundness of specimens.

The ultrasonic inspection technique used throughout this program is to monitor the acoustic amplitude corresponding to a reflection thickness resonance of the back surface echo and use this amplitude to modulate the intensity of a C-scan, reference (c). Using this technique, it was possible to detect delaminations in the laminates as small as 1/8 in. (3 mm) in diameter occurring at any depth throughout the thickness. Figure 1 presents a C-scan for an undamaged plate and figure 2 shows typical delamination damage. Although the C-scan does not record the depth of the damage, the equipment operator can read damage depth from an oscilloscope display of the ultrasonic pulse-time response when this information is needed.

STATIC TESTS

Reference (d) found that for low speed central impact on composite beams, the dynamic failure mechanism was identical to the static failure mechanism, in that case longtudinal tension failures of the 0^{o} plies with the dynamic failure strain equal to the static failure strain. Wave effects were not important in the velocity range studied, up to 20 ft/sec (6.1 m/s), and the impact response was primarily a structural dynamic phenomena. Assuming the behavior of plates will follow the same pattern, a static test was performed to serve as a basis with which to compare impact tests.

Two specimens of laminate B were tested. Each specimen was supported along all edges as a simply supported plate in a rigid frame and a steel indenter was pressed into the center of the plate, figure 3, until damage resulted. Here damage was determined by a lound sound in company with a sudden drop in the load. One test used a 1/4 in. (6.4 mm) radius indenter and the other a 1 in. (25.4 mm) radius indenter. A strain rosette was located on the back face of the specimen opposite the contact point. Figure 4 is a record of the load versus strain during this test. Note that although the flexural response of the two indenters is identical, i.e., both ε_x and ε_y follow the same strain versus normal load curve, the failure loads are different by 37 percent. Results of visual and ultrasonic inspection are given in table II.

TABLE II

STATIC TEST RESULTS

Indenter Radius	Visual Damage	NDE Damage
1/4 inch	None	Elliptical mid-plane delamination centered under contact point Length = 1.15 inch Width = 1.25 inch
1 inch	None	Elliptical mid-plane delamination centered under contact point Length = 2.05 inch Width = 2.63 inch

The damage mode resulting from a transverse normal static load on this geometry specimen is clearly a transverse shear failure. The smaller the radius of the indenter, the smaller the normal load required to cause damage. This is due to the larger local transverse shear stress caused by the smaller indenter

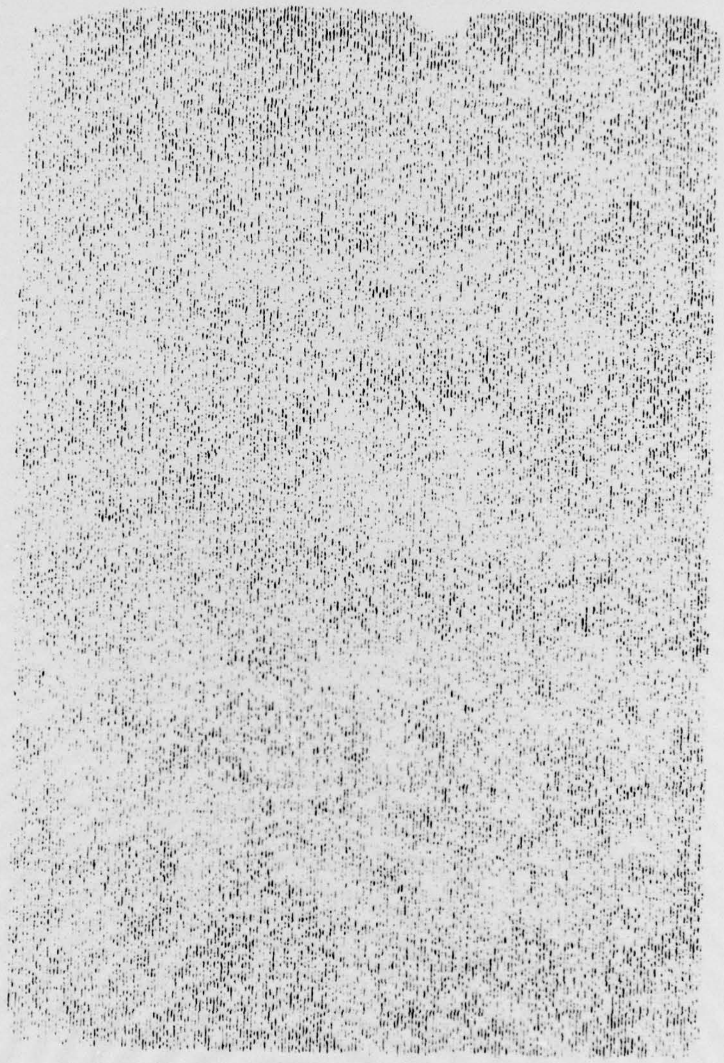


FIGURE 1 - C-Scan of Undamaged Plate.

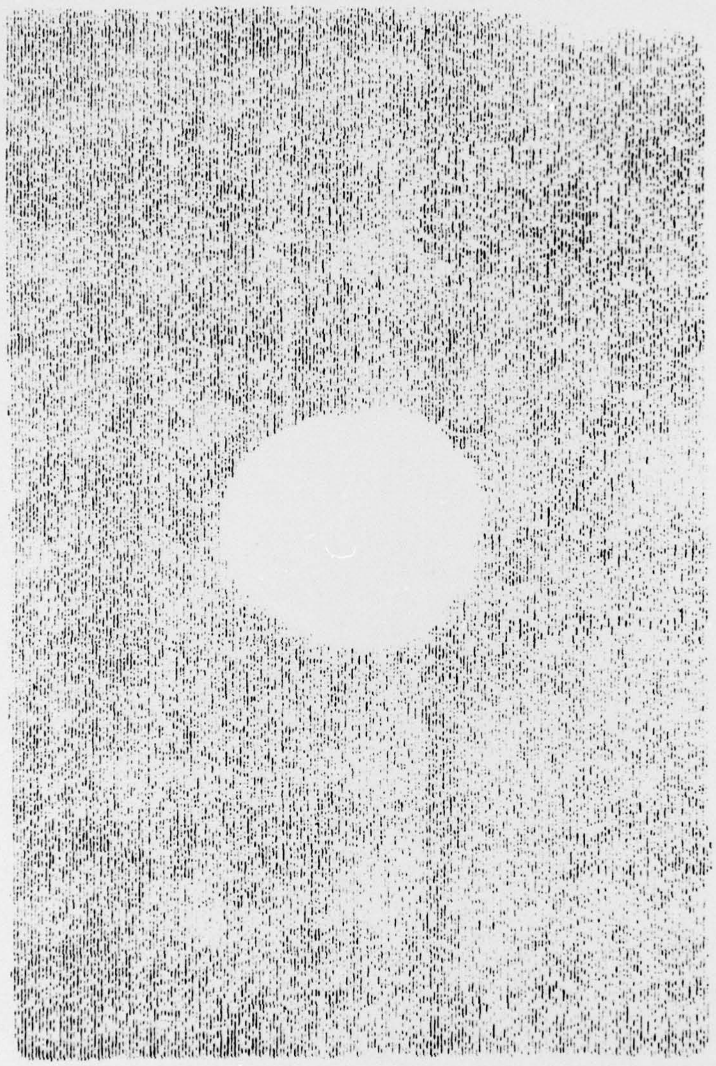


FIGURE 2 - C-Scan of Typical Delamination Damage.

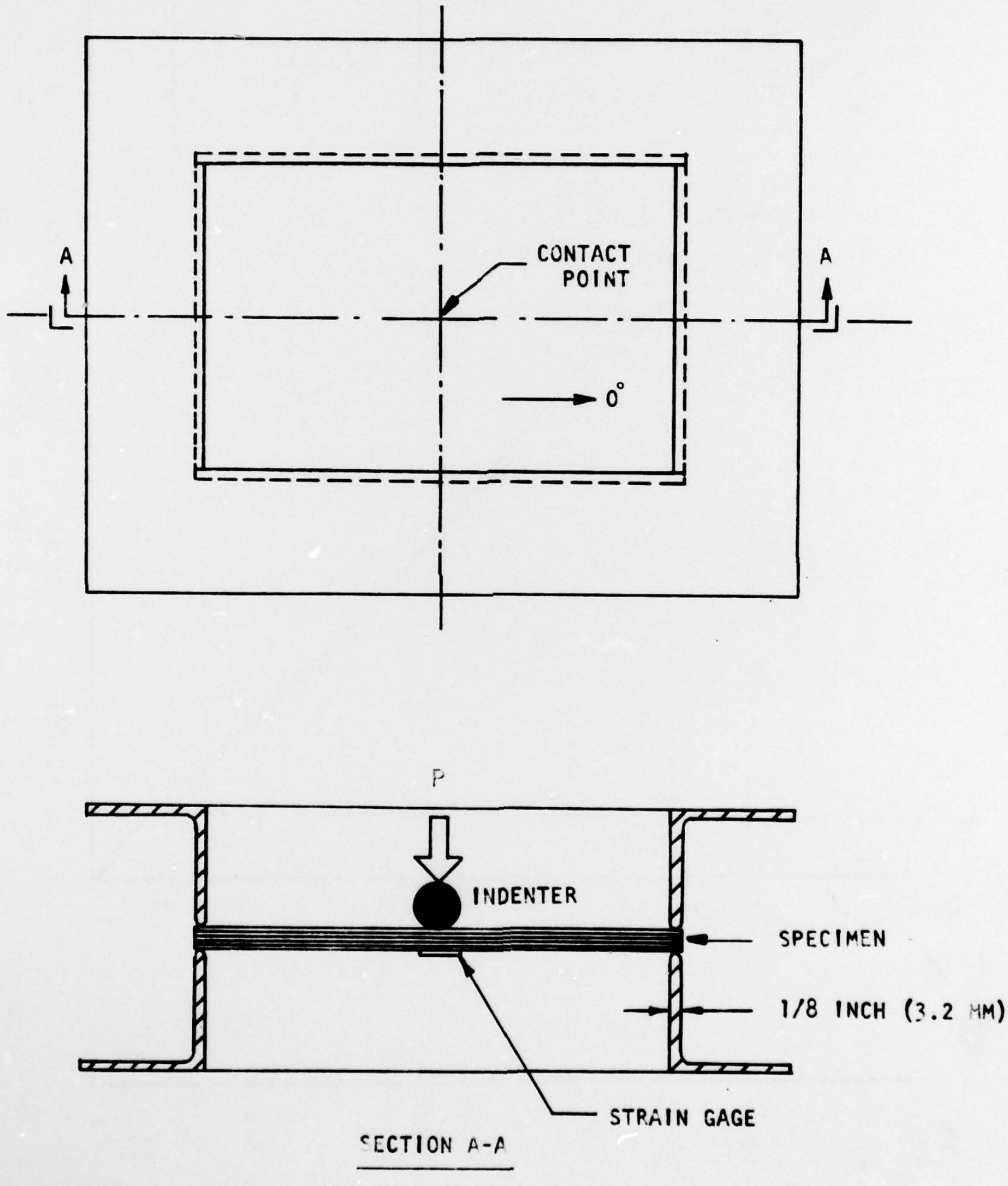


FIGURE 3 - Static Test Set-Up.

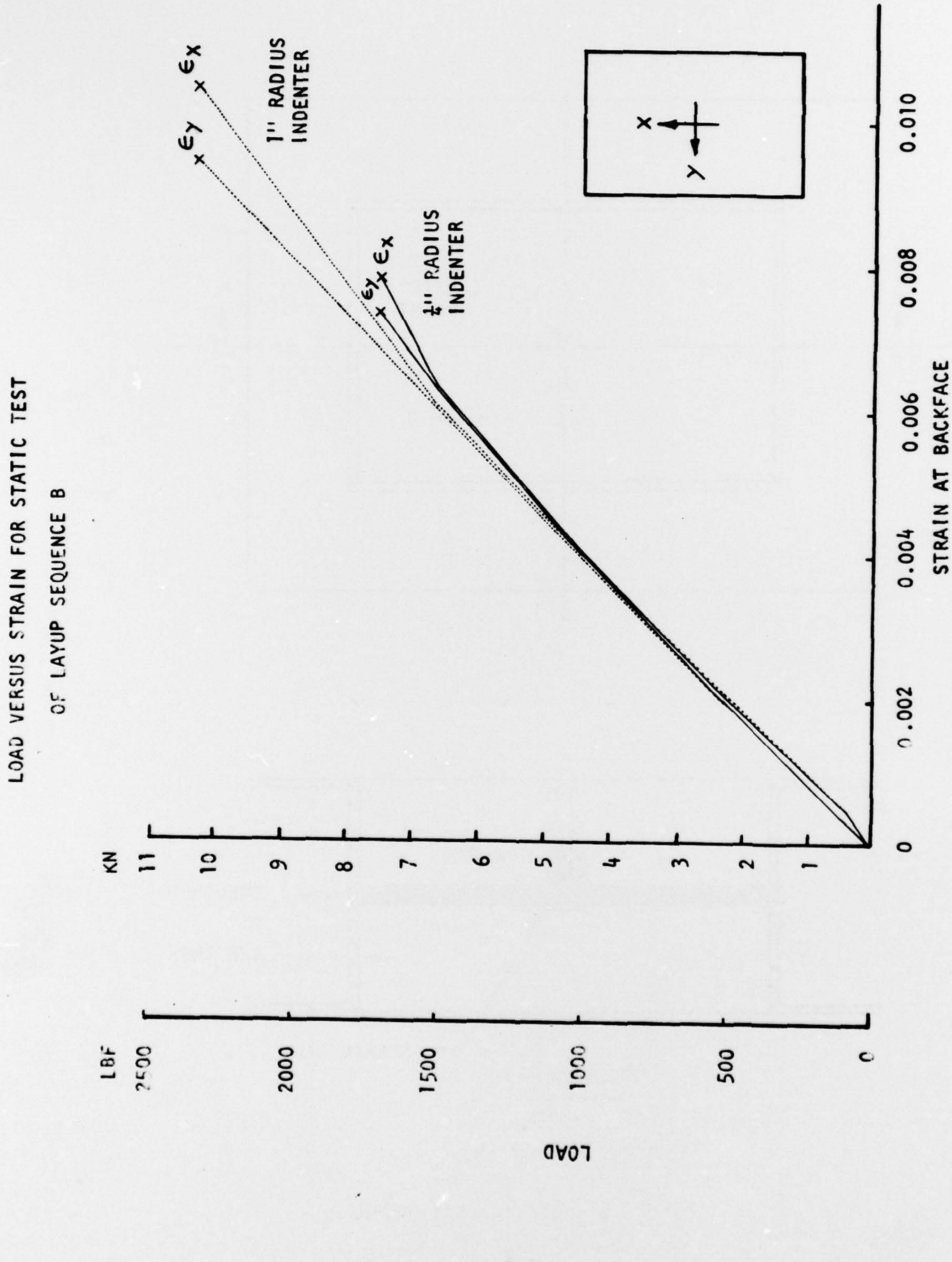


FIGURE 4 - Static Test Load vs Backface Strain.

radius. The damage mode would be expected to shift from a shear to bending mode with an adequate decrease in the thickness or increase in the size (length and width) of the panel, similar to that observed in beams. When the large radius indenter is loaded sufficiently to cause damage, the resulting damage area is greater due to the increased energy (as a result of the greater load) which must be dissipated in the creation of fracture surfaces following failure.

IMPACT TESTING

During impact tests, each specimen was supported along its edges as a simply supported plate in the same frame used during the static tests. A steel indenter of 1/4 in. (6.4 mm) tip radius and variable mass was employed to simulate a typical tool. The indenter was dropped from various heights so that it struck the specimen once at its center and was caught on rebound.

An impact damage threshold was determined for each layup sequence by incrementally increasing the drop height of a 1 lbm (454 g) indenter until damage could be detected either visually or by ultrasonic inspection. Damage threshold was also determiend for laminate B using a 2 lbm (908 g) indenter. Additional impacts were conducted above threshold level to examine impact threat versus damage magnitude relations. Results are presented in table III and figures 5 and 6.

The impact damage mode was similar to the static test damage mode, i.e., transverse shear type failures. But unlike the static test, where incipient damage was a mid-plane delamination of approximately 1 inch diameter, incipient impact damage was slight delamination ($\sim$1/16 in. (1.59 mm) diameter) located approximately 10 percent depth below the impact point. By varying the impact parameters of mass and velocity, it was possible to induce different levels of damage from just perceptible to large (approximately 4 in. (102 mm) diameter) delaminations with obvious front and back surface damage.

Figure 7 is a plot of damage area versus impact energy. The laminates experience no effects from impacts up to approximately 5 ft-lbf (6.8J) when the slight incipient damage occurs. Increased energy impacts result in greater damage. Experimental scatter in damage areas, particularly the 8.22 ft-lbf (11.1J) impacts on B laminates, was extreme. An examination of the location in the original panel from which the specimens were extracted showed that the specimen possessing the greatest impact resistance was taken from the central portion of that panel and was thus the most uniform in thickness and properties, and the specimen suffering the greatest damage was extracted from a corner of the original panel. These results accent the need for careful layup and cure of advanced composite materials to ensure uniformity of properties and high quality for the finished part.

In addition to variations in part quality, the size of the damage area can vary according to the nature of the fracture surfaces and extent of cracking within a given volume. Because no photomicrographs were made of these specimens, the actual fracture surface areas could not be approximated. Damage extent based on actual fracture surface area may correlate better than gross C-scan detectable area.

TABLE III

IMPACT DAMAGE MAGNITUDE

1/4 Inch Radius Indenter

Specimen No.	Impact Mass (LBM)	Drop Height (Ft)	Visual	C-Scan (Length x Width) In.	Impa V = √
A1	1.01	1.00	None	None	
A2	"	2.00	"	"	
A2	"	3.50	"	"	
A1	"	4.00	"	"	
A2	"	6.00	"	Delamination below impact point between 4th and 5th ply ¼ inch diameter	
A1	2.01	"	Slight back face cracking	(1.63 x 1.56) see figure 5	
A2	"	"	"	(1.81 x 1.50)	
A3	"	"	"	(1.65 x 1.40)	
A4	"	"	Slight front face cracking	(1.75 x 1.30)	
A5	"	"	Slight back face cracking	(1.70 x 1.55)	
A6	"	"	"	(1.85 x 1.55)	
B1	1.00	3.73	None	None	
B1	"	4.11	"	Delamination at 10% depth 1/16 inch diameter	
B2	"	4.85	"	(0.20 x 0.14)	
B1	"	5.61	"	(0.50 x 0.48)	
B3	2.00	1.50	"	None	
B3	"	1.63	"	"	
B3	"	1.75	"	"	
B3	"	2.00	"	Delamination at 10% depth 1/16 inch diameter	
B1	"	4.11	"	(2.06 x 2.15)	
B4	"	"	"	(0.10 x 0.12)	
B5	"	"	"	(1.30 x 1.38)	
B6	"	"	"	(1.30 x 1.45)	
B7	"	"	"	(1.70 x 1.90)	
B8	"	"	"	(1.50 x 1.75)	
B9	"	6.00	Slight back face cracking	(1.64 x 1.95)	
B10	"	"	"	(1.68 x 1.95)	
B4	3.00	"	Detectable front face dent, back face cracking, see figure 6	(1.80 x 2.14)	
B2	"	"	Detectable front face dent, back face cracking, see figure 6	(2.02 x 2.54)	

TABLE III

IMPACT DAMAGE MAGNITUDE

1/4 Inch Radius Indenter

C-Scan (Length x Width) In.	Calculated Data: Impact Velocity $V = \sqrt{2gh}$ (ft/sec)	Kinetic Energy $1/2\ mv^2$ (ft-lb)	C-Scan Damage Area $\frac{\pi}{4}$ AxB (m^2)	
None	8.02	1.01	None	
"	11.35	2.02	"	
"	15.01	3.54	"	
"	16.05	4.04	"	
Delamination below impact point between 4th and 5th ply ¼ inch diameter	19.66	6.06	0.05	
(1.63 x 1.56) see figure 5	"	12.06	2.00	
(1.81 x 1.50)	"	"	2.13	
(1.65 x 1.40)	"	"	1.81	2.01 Avg.
(1.75 x 1.30)	"	"	1.79	
(1.70 x 1.55)	"	"	2.07	
(1.85 x 1.55)	"	"	2.25	
None	15.50	3.73	None	
Delamination at 10% depth 1/16 inch diameter	16.27	4.11	0.003	
(0.20 x 0.14)	17.67	4.85	0.02	
(0.50 x 0.48)	19.01	5.61	0.19	
None	9.83	3.00	None	
"	10.25	3.26	"	
"	10.62	3.50	"	
Delamination at 10% depth 1/16 inch diameter	11.35	4.00	0.003	
(2.06 x 2.15)	16.27	8.22	3.48	
(0.10 x 0.12)	"	"	0.01	
(1.30 x 1.38)	"	"	1.41	1.83 Avg.
(1.30 x 1.45)	"	"	1.48	
(1.70 x 1.90)	"	"	2.54	
(1.50 x 1.75)	"	"	2.06	
(1.64 x 1.95)	19.66	12.00	2.51	2.54 Avg.
(1.68 x 1.95)	"	"	2.57	
(1.80 x 2.14)	"	18.00	3.03	3.53 Avg.
	"		4.03	
(2.02 x 2.54)	"	"		

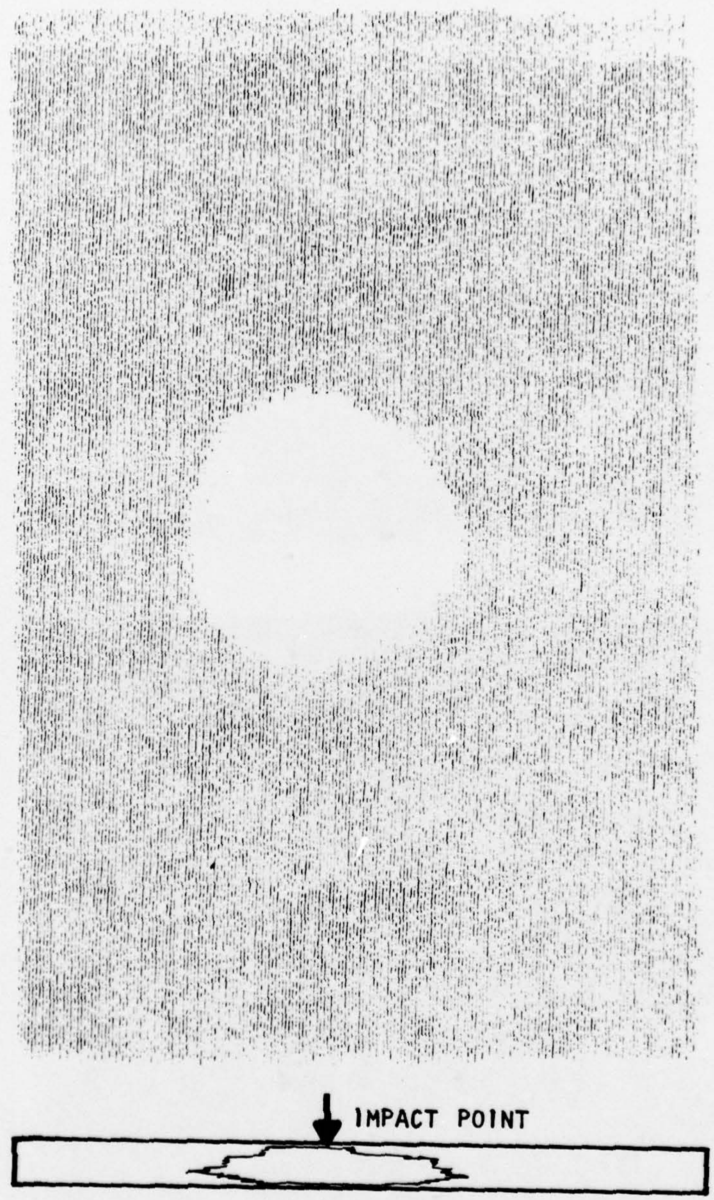


FIGURE 5 - Typical C-Scan of Impact Damage. (48 ply 4 inch x 6 inch simply supported plate impacted by 1/2 inch diameter contactor of 2 LBM dropped from 6.0 feet.)

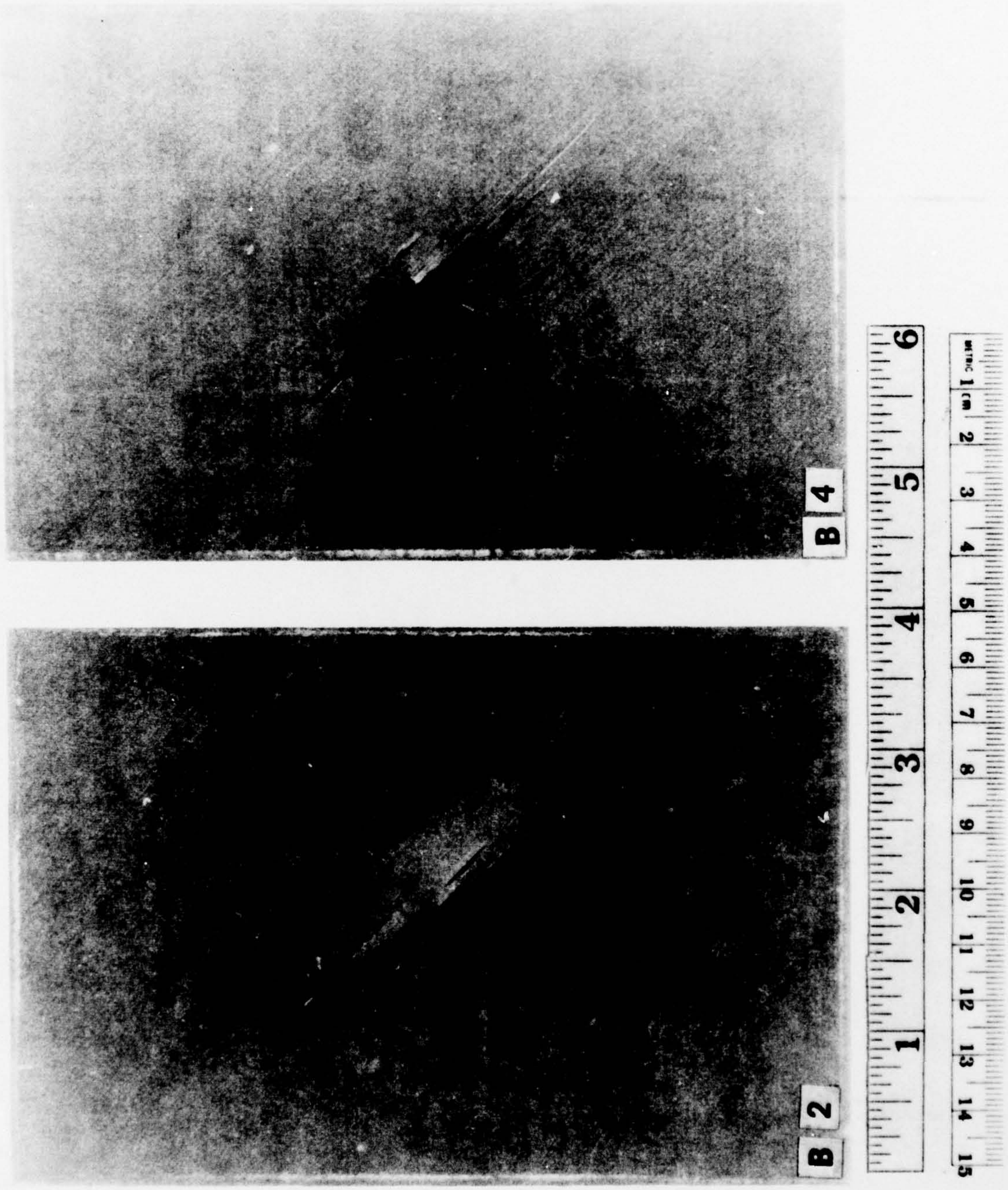


FIGURE 6 - Backface Visual Damage. (Sequence B layup impacted by 3.00 LBM, 1/2 inch diameter weight dropped rom 6 feet.)

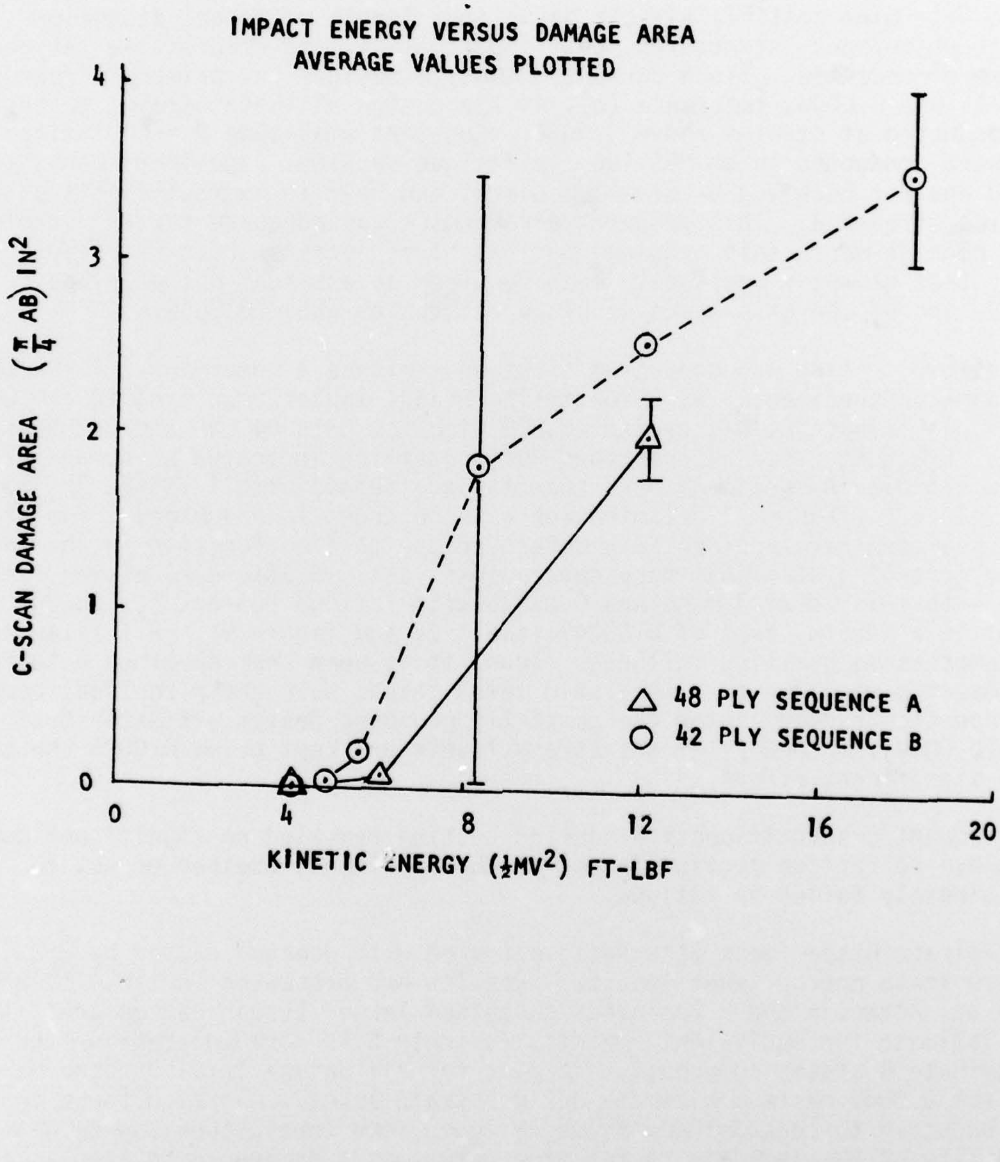


FIGURE 7 - Impact Engery vs Damage Area. (1/2 inch tip diameter dropped weight test.)

Examination of the impact energy versus damage area for both laminate A and B indicates laminate A to sustain less damage than laminate B. In both laminates impacts less than 12.0 ft-lb induced no visible front face damage with back face damage apparent at 12.0 ft-lb impacts. In laminate B the 18.0 ft-lb impact resulted in an obvious (but small) dent at the impact point with extensive back face cracking.

POST IMPACT COMPRESSION TESTS

To determine possible effects of various levels of impact damage on thick graphite/epoxy structures, post-impact static and compressive fatigue tests were conducted. Since current composite designs for primary structure typically use 0.0040, reference (e), as the design ultimate strain, no tests were conducted at strains above 0.0050. Constant amplitude R =-00 fatigue tests were conducted in an MTS 100 kip fatigue machine. Specimens were stabilized against buckling by aluminum plates and held in hydraulic self-aligning grips, figure 8. This gripping arrangement was adequate for the strain levels considered in this program, but had higher strains been required, another test geometry would have been required as a slight out of plane displacement of the grip heads develops at strains above 0.0056.

Fatigue testing was conducted first on laminate A specimens. A threshold level damaged specimen, i.e., 6.06 ft-lb (8.15J) impact resulting in a 0.05 in.^{2} (32 mm^2) delamination, was cycled 10,000 times at both 0.0040 and 0.0050 strain levels. Frequent C-scan inspections during testing indicated no damage growth. All six laminate A specimens were than fatigue tested with 12.06 ft-lb impact damage (2 in.^{2} (1290 mm^2) delamination with no front face damage). Two of these specimens were accidentally destroyed due to a malfunction in the fatigue machine control system and these data points lost. Static compression failure strain with this level damage was 0.00378 with Fatigue run-out (2,000,000 + cycles) at a strain level of 0.00243 (table IV and figure 9). All failures were compressive buckling failures. Thus, it is seen that specimen A laminates with impact damage due to a realistic level threat have their residual compressive properties reduced below the currently proposed design ultimate strain level (0.0040), although, if the strain levels are kept below 0.0025 the damage has no significant effect.

Frequent C-scan inspections during testing revealed no significant damage growth due to fatigue cycling in any of the specimens, whether or not the specimen ultimately failed in fatigue.

Laminate B specimens were fatigue tested with damaged caused by 8.22, 12.0, and 18.0 ft-lb energy level impacts. Results are presented in table IV and figure 9. Although the B laminates sustained larger C-scan damage areas than the A laminate for equivalent impacts, laminate B is more tolerant of the damage. The laminate B static compressive strains for all damage levels tested were above the 0.0040 maximum ultimate design strain level. Although these tests were conducted to constant amplitude fatigue, some conclusions can be made about tne ability of the laminate to function with impact damage in an actual aircraft structure. If we assume that the low load cycles can be truncated in the flight by flight NAVAIR fatigue spectrum used in the structural validation testing of the AV-8B composite wing design, (a very severe spectrum) and that the only number of maximum load cycles are important to the fatigue life of the laminate,

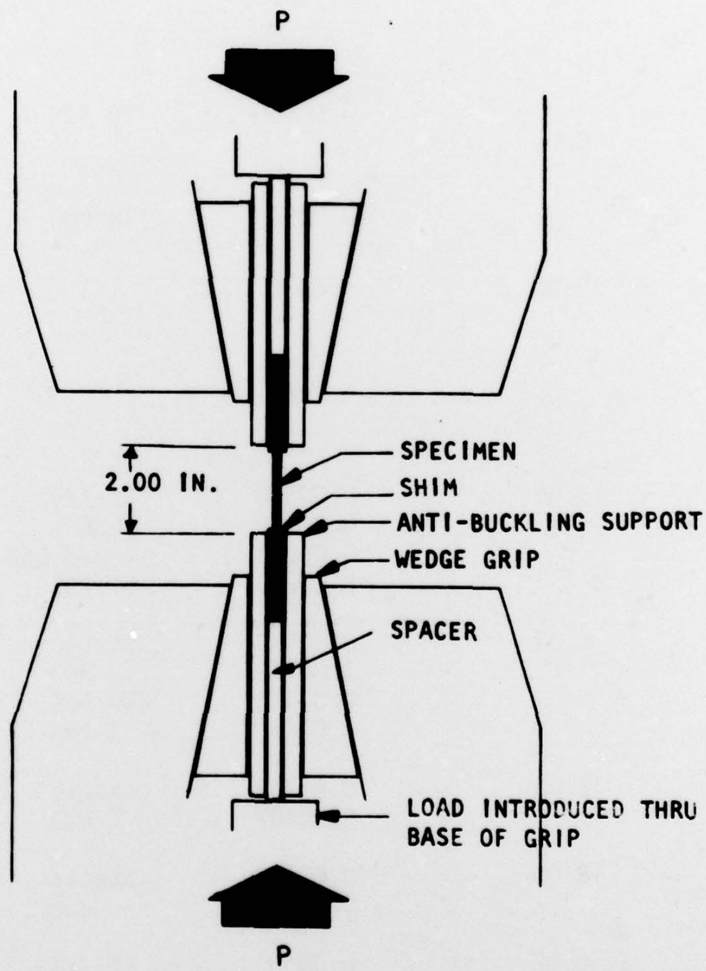


FIGURE 8 - Compression Fatigue Test Grip Arrangement.

TABLE IV

STATIC AND FATIGUE TEST RESULTS

Specimen	Impact Energy 1/2 mv^2 (ft-lb)	Maximum Gross Compressive Strain	Cycles to Failure (at 1.5 Hz)	Comments
A5	None	0.00559	Static +* **	No Effect - Validate Test Fixture
A2	6.06	0.00400	10 900 +	No Effect - Retested at Highest Strain
		0.0050	10 000 +	No Effect
A1 A2	12.06	<.0040		Specimens Accidentally Destroyed due to Fatigue Machine Malfunction
A3	12.06	0.00378	Static	
A4	"	0.00300	1836	Slight Damage
A5	"	0.00243	2 000 000 +	Growth (<1/16 in.)
A6	"	0.00270	61 255	
B5	8.22	0.00482	Static	See figure 10
B6	"	0.00452	3 069	
B7	"	0.00438	496 509	
B8	"	0.00465	2 184	
B9	12.00	0.00463	Static	
B10	"	0.00407	533	
B4	18.00	0.00417	Static	
B2	"	0.00367	655	
B1 Repaired	8.22	0.00500	26 417	See figure 11

* Static indicates single, gradually increasing load cycle.

** A plus sign at end of entry indicates specimen did not fail.

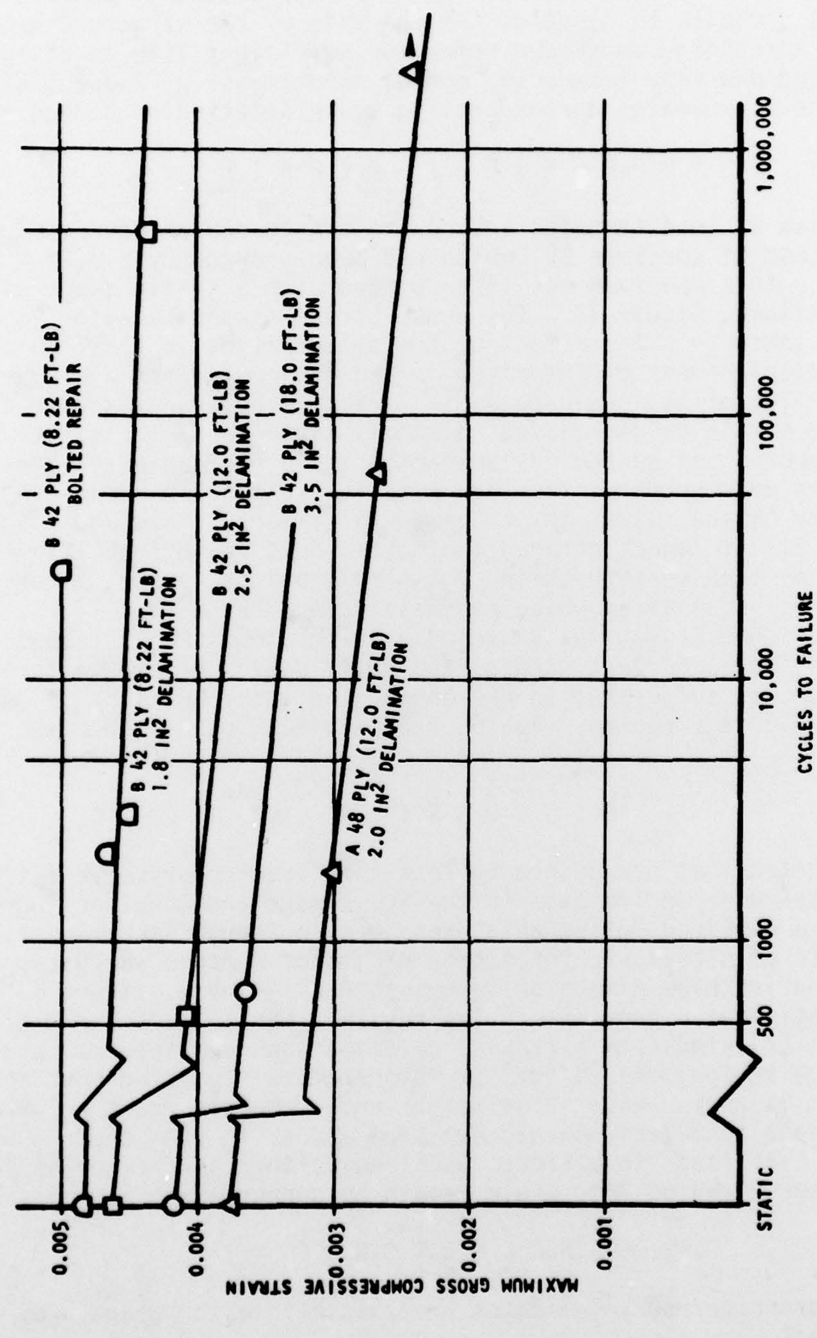


FIGURE 9 - Life to Failure.

then the laminate would only need to withstand the 0.004 strain level 1000 times during one aircraft lifetime. Keeping in mind possible errors associated with the truncation of the low cycle loads, it appears that the B laminate can sustain visually undetectable impact damage and still possess sufficient residual strength to function for the life of the structure without repair. Visual threshold damage can serve for sufficient life to allow the damage to be found and repaired during normal maintenance. Frequent C-scan inspection of the B laminates during cycling again detected no damage growth.

SIMPLE BOLT REPAIR

The mechanism of load transfer around the damage region was examined during the static test of specimen B5, which had been damaged by a 8.22 ft-lb (11.14J) impact. This specimen was instrumented with a strain gauge array and loaded to failure, figure 10. The gross stress-strain behavior is the same as an undamaged laminate. The effect of the delamination is to allow transverse (out-of-plane) displacement of the plies in the damaged region and therefore allow local buckling of the laminate as is indicated by appearance of the failed specimen and the strain gauge readings directly over the impact point. It appears that the sub-visual damage (delaminations) did not damage the graphite fibers. Failures under compressive load results from matrix damage and the resultant inability of the matrix to stabilize the fibers. This would suggest that a possible fix of impact damaged laminates would be to stabilize the laminate against local transverse buckling. A simple bolted repair, figure 11, was tried on a laminate B specimen which had been damaged by an 8.22 ft-lb impact. This specimen was then successfully cycled at a strain of 0.0050 (20 percent greater than the static failure strain) for 26417 cycles before failure, indicating the repair was successful in preventing the local buckling of the graphite fibers in the damage region. Again, frequent NDI inspection revealed no damage growth.

DISCUSSION

It was expected that damage growth resulting from compressive fatigue cycling would have been an increase in the NDI damage area, but no increase in damage areas were observed during this program even though specimens failed in fatigue. Results of ultrasonic inspection of impact damaged laminates indicate the damage region is three dimensional through the laminate, figure 5, but gives no information of damage within the region. Photo micrographs of impact areas, reference (f), indicate extensive delaminations and internal cracking inside the damage region, figure 12. It is therefore suggested that the damage growth mechanism is an increase in delamination and matrix cracking within the original NDI damage zone (which would not then appear to grow under subsequent NDI inspection) that leads to ultimate buckling failure in fatigue as the fibers in the damage region lose their remaining support from the matrix.

CONCLUSIONS

1. Thick graphite/epoxy laminates are susceptible to damage under realistic impact threats.

2. The dominate damage mechanism of thick composite plates under transverse normal impact loading (up to 20 ft/sec) is a static type shear failure mode and wave effects are not important.

STATIC COMPRESSIVE STRAIN TO
FAILURE FOR 42 PLY LAMINATE DAMAGED BY
8.22 FT-LB IMPACT.

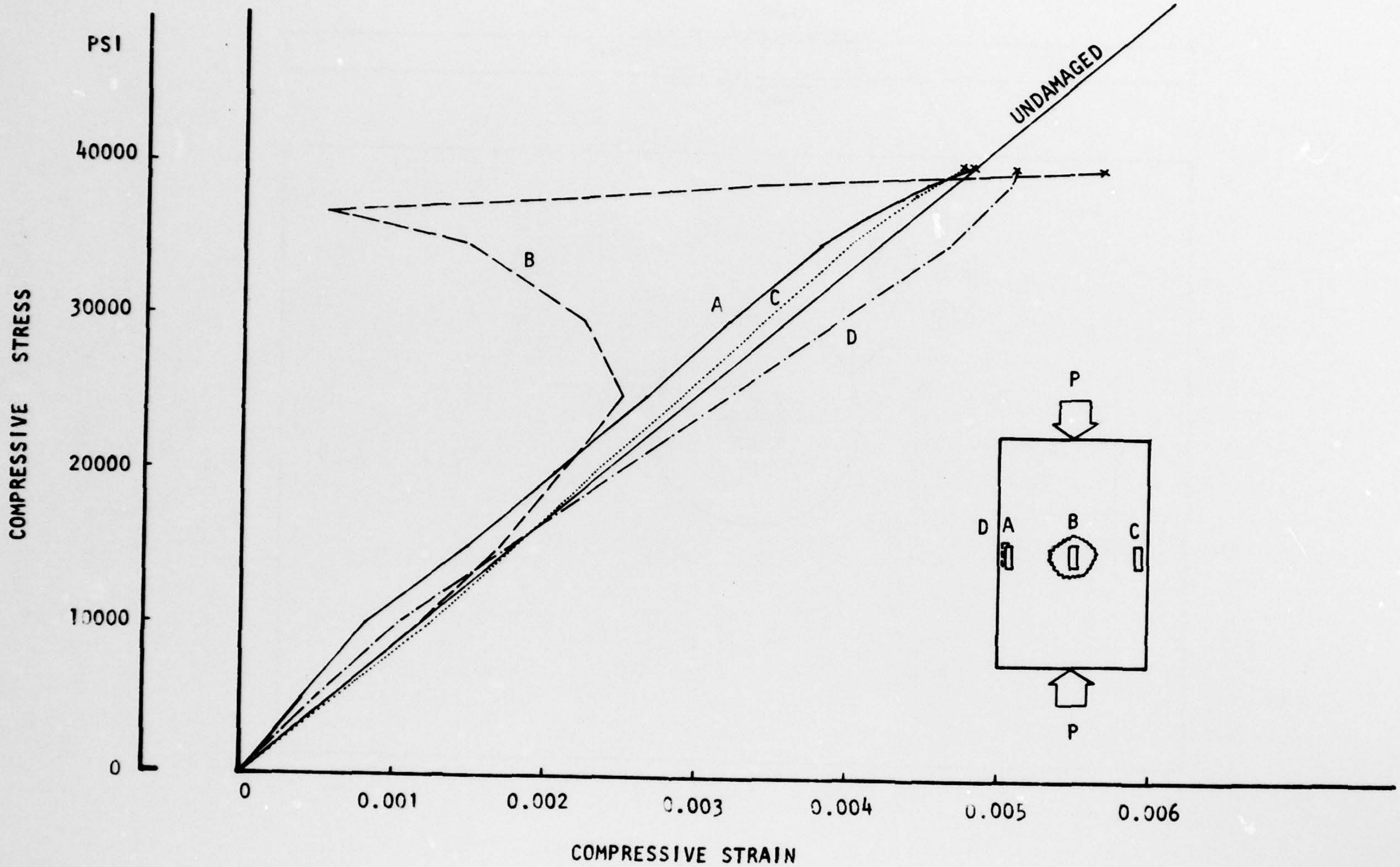


FIGURE 10 - Static Test of Damaged Specimen.

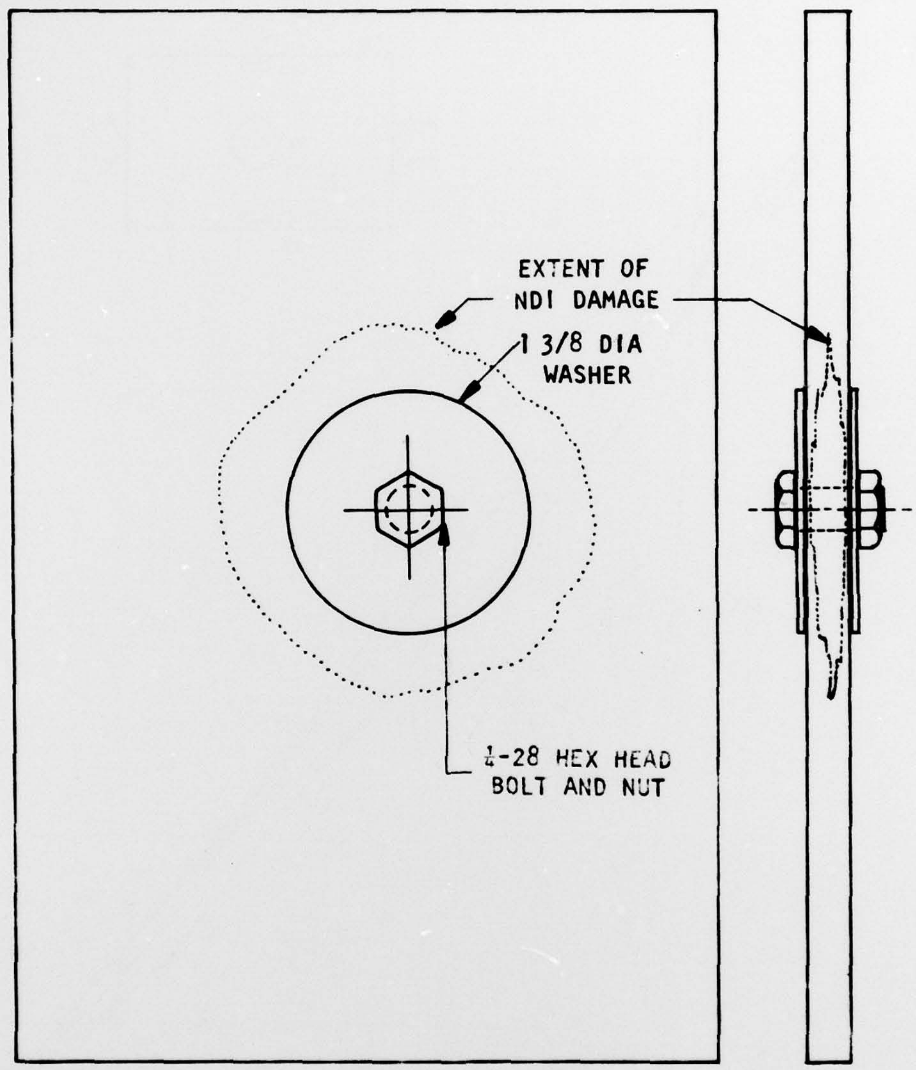
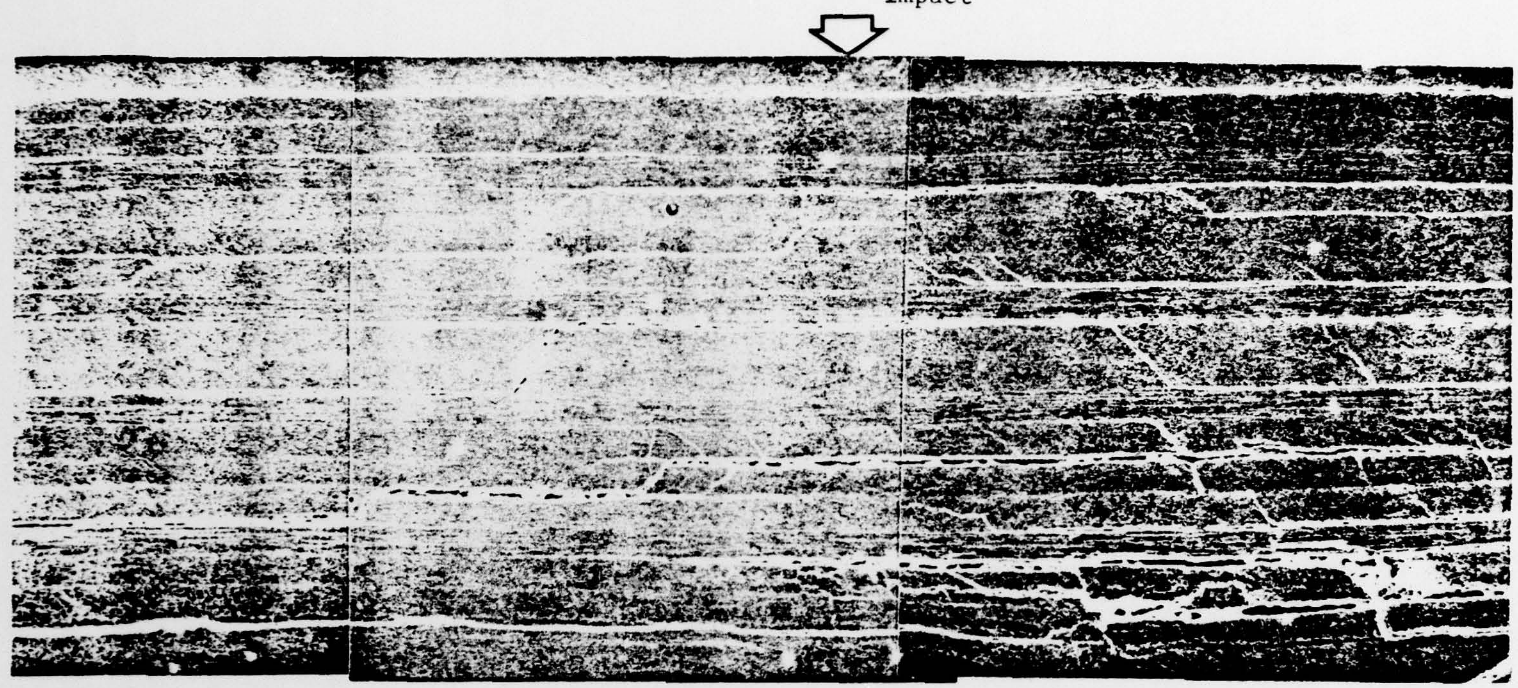


FIGURE 11 - Simple Bolted Repair.



Panel 901-5, Location D
16 Plies Gr/Ep $[(\pm 45/0/90)_2]_s$

Blunt Impactor at Center of 5 inch Square Area
Total Absorbed Energy = 2.41 ft. lbs.
(Incipient Damage Indicated at 1.93 ft. lbs)
Damage Not Visible on Impacted Surface
Slight Matrix Crack on Back Face

FIGURE 12 - Photomicrograph of Impact Area of 16 Ply Panel 901-5 (D)/
(From reference (f)).

3. Failure of impact damaged thick laminates under axial compressive loading is due to local buckling of the graphite fibers resulting from the inability of the matrix to stabilize the fibers as a consequence of shear failure of the matrix resulting from impact.

4. Laminate construction is an important parameter in influencing damage tolerance. Impact damage on 49-ply $(\pm 45/0_2/\pm 45/0_2/\pm 45/0/90)_{2S}$ laminates which was not visible reduced the performance below current composite design allowables (design ultimate strain 0.0040), although the laminate could tolerate damage at strain levels less than 0.0025. Forty two ply $(+45/90/-45/+22.5/-67.5/-22.5/+67.5/\pm 45/+67.5/+22.5/-67.5/-22.5/\pm 67.5/\pm 22.5/0_2/\pm 22.5)_S$ laminates can tolerate subvisual impact damage at current design allowables.

5. Indenter diameter is an important parameter influencing damage.

6. For the particular laminates and impact geometry studied, there is no damage for impact energy less than 4.0 ft-lb, damage becomes visible on the back face at approximately 12.0 ft-lb impact energy, and damage is visible on the front face at 18.0 ft-lb impact energy.

RECOMMENDATIONS

1. An analytical effort should be directed to determining the shear stresses in thick laminates resulting from impact.

2. Reduced strain allowables should be used in design of graphite/epoxy structures. In the case of the laminate B construction studied here, the current design ultimate strain of 0.0040 is low enough to allow a structure with invisible impact damage to fulfill its design life with no repair.

3. Additional tests should be made to verify the results of this exploratory program and provide a statistically valid data base. Included should be photomicrographic studies on similarly damaged specimens sectioned after various numbers of fatigue cycling to determine the fatigue damage growth mechanism.

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